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Standard Tests for Toughened Resin Composites

Revised Edition

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Standard Tests for Toughened Resin Composites

Revised Edition

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Scientific and Technical
Information Branch

FOREWORD

Technology leading to the application of advanced composites in secondary and empennage structures for commercial transport aircraft has been under development in the Aircraft Energy Efficiency (ACEE) Program for several years. This program has achieved a high degree of success; the aircraft manufacturers are now in a position to consider composite versions of such structures in plans for future aircraft. Their consideration will be backed by personnel experienced in all areas of design, manufacture, and quality control; by a tested interface with material suppliers, the FAA, and the airlines; and by first-hand experience with the cost and weight benefits of such composite structures.

Although the application of composites to secondary and medium primary components can achieve a fuel savings of up to 3 percent on current transport aircraft, studies indicate that weight savings associated with extensive use of composite structures in wing and fuselage components will lead to a fuel savings of 10 to 15 percent over similar aircraft with conventional structures. Opportunities for technology advancements, which will significantly enhance these benefits, exist in areas such as tougher resin development, thick laminate processibility, durability and damage tolerance, and strength-critical design optimization. In fact, research programs are underway within NASA and the Aircraft Industry to develop resin matrix composites which will exploit the full weight-reduction potential for strength-critical primary aircraft structures. The immediate goal in these programs is to increase the allowable design strain to 6000 μin/in. while maintaining desirable features in processibility, mechanical properties, and environmental stability.

Concurrent with the new material development, the commercial transport manufacturers are under contract to NASA to develop long-lead-time technology for application of advanced composites to primary aircraft structures. Since several toughened resin systems are being evaluated in these contracts, NASA conducted a workshop in December 1981 to achieve commonality among the manufacturers for certain kinds of tests used to characterize toughened resin composites. Out of this workshop, specifications for five tests were standardized and described in NASA Reference Publication 1092, May 1982. Tests of double cantilever beam specimens resulted in an excessive amount of scatter in the data. The scatter was attributed to the specimen not providing a pure Mode I condition. To correct this situation, the hinged double cantilever beam specimen will replace the double cantilever beam specimen. A description of the revised test standard ST-5 is given in this revised edition of RP-1092. In addition, data tables have been added to insure uniform reporting of test results. For more detailed information on such items as apparatus and procedures, the reader should contact the principal contributor listed in the table on the following page.

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STANDARD TESTS FOR TOUGHENED RESIN COMPOSITES

Test designator	Test type	Ply orientation	Nominal thickness	Test specimen size, in.	Principal contributor
ST-1	Compression after impact	$[+45/0/-45/90]_{ns}$	0.25 in.	7 by 12.5 (before impact) 5 by 12.5 (after impact)	Charles F. Griffin Lockheed-California Company Burbank, California 213-847-3862
ST-2	Edge delamination tension	$[\pm 30/\pm 30/90/\overline{90}]_s$ $[\pm 35/0/90]_s$	11 plies 8 plies	1.5 by 10	T. Kevin O'Brien Structures Laboratory AVRADCOM Research and Technology Laboratories NASA Langley Research Center Hampton, Virginia 804-865-3011
ST-3	Open-hole tension	$[+45/0/-45/90]_{ns}$	0.25 in.	2 by 12	D. J. Watts Douglas Aircraft Company Long Beach, California 213-593-3442
ST-4	Open-hole compression	$[+45/0/-45/90]_{ns}$	0.25 in.	5 by 12.5	Charles F. Griffin Lockheed-California Company Burbank, California 213-847-3862
ST-5	Hinged double cantilever beam	$[0]_n$	0.12 in.	1.5 by 9	Motoaki Ashizawa Douglas Aircraft Company Long Beach, California 213-593-2126

ST-1: SPECIFICATION FOR COMPRESSION AFTER IMPACT TEST

INTRODUCTION

The ST-1 specification defines the test specimens, test apparatus, test procedures, and data to be compiled for compression tests on graphite/epoxy laminates after impact.

DESCRIPTION OF TEST LAMINATE

The graphite/epoxy test laminate shall have a nominal thickness of 0.25 in. and an orientation of $[+45/0/-45/90]_{ns}$. The laminate shall be cured with a 0.25-in.-diameter disc made of Du Pont Teflon or equivalent (1 mil thick) embedded in one corner for an NDI (nondestructive inspection) standard and shall be ultrasonically inspected to determine laminate quality and to provide a basis of comparison for postimpact ultrasonic inspection. Record laminate thickness and resin content. Place an identification number on one side of the laminate. Do not paint the laminate.

TEST SPECIMEN DIMENSIONS

The impact test specimen (fig. 1) shall have a width of 7.00 in. The length shall be not less than 10.0 in. nor greater than 12.5 in. After impact, the specimens shall be trimmed to a width of 5 ± 0.03 in. for compression test to failure.

TEST APPARATUS

Impact Test

The impact test apparatus (fig. 2) shall consist of a base plate, a top plate, and an impactor. The impactor shall weigh 10.0 lb, be less than 10 in. in length, and have a 0.5-in. hemispherical steel tip on the end that impacts the specimen. A guide tube lined with Du Pont Teflon or equivalent for the impactor is also required for minimum friction.

Compression Test

The compression test apparatus (fig. 3) shall provide simple support to the compression test specimen along its long edges oriented parallel to the compression loading direction. The short edges (loaded edge) shall be clamped between the two adjustable steel plates of the upper and lower sections of the apparatus to provide resistance to end brooming.

TEST PROCEDURE

Impact Test

The graphite/epoxy test specimen shall be placed in the impact test assembly (fig. 2) with the identification side up so that the desired impact location is centered within the 5.0- by 5.0-in. central opening in the base plate. The top plate shall then be placed upon the test specimen and clamped to the base plate by installing nuts on the four tie-down studs and torquing each one to a nominal 20 ft-lb. Alignment pins are provided in the base plate so that the top plate is correctly positioned. The guide tube shall be located above the test specimen so that the impactor will strike the center of the specimen. The lower end of the guide tube shall be approximately 10 in. above the surface of the specimen. Coat the striker end of the impactor with white chalk dust or white grease to allow easy location of the actual impact point. Drop the impact from a height of 2 ft above the test specimen to generate an impact energy of 20 ft-lb. Care should be taken to deflect the impactor away from the specimen after the strike so that a restrike does not occur. Remove the impacted specimen from the test apparatus and visually (naked eye) determine the amount of damage to the specimen on the impacted surface (side with the specimen identification) and the back surface. The specimen shall then be ultrasonically inspected to determine the extent of internal delamination.

Compression Test

The compression test specimen shall be loaded to failure by using a stroke-controlled testing machine. A loading rate of 0.05 in/min is recommended. The load-strain behavior of the test specimen shall be recorded throughout the test by using strain gages. The specimen shall be installed in the compression test apparatus (fig. 3) such that (1) the specimen is parallel to the load axis of the machine and is centered in the machine; (2) the side supports on the edges parallel to the loading axis shall be a snug fit, but not tight, so that the specimen can still slide in the vertical direction; and (3) a 0.050-in. clearance is provided between each side of the specimen and the side supports to prevent any transverse load due to Poisson deformation being restrained.

NUMBER OF TESTS

Three impact tests shall be conducted at an impact energy of 20 ft-lb. These three specimens shall be tested to failure in compression.

TEST DATA REPORTING

For each impact test, record the data as specified in table 1 where h is the thickness of one ply of the laminate. Also provide ultrasonic indication measurements using C-scan and, where possible, associated B-scan records.

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TABLE 1.- ST-1 COMPRESSION AFTER IMPACT TEST DATA

Company affiliation: ______________________

Material: ______________ h = ______ mils/ply

Laminate orientation: $[+45/0/-45/90]_{ns}$
Resin content: ___% by weight
Test condition: 75°F dry

Specimen ID	Thickness, in.	Width, in.	Impact energy, ft-lb	Maximum width of impact damage, in.	Visual impact damage		Impact area, in^2	Failure load, kips	Failure stress, ksi	Failure strain, μin/in.	Compression modulus, psi
					Front surface	Back surface					
	x.xxxx	x.xxx	xx.	x.xx	Yes	No	x.xx	xx.xx	x.xx	xxxxx.	$x.xx \times 10^6$

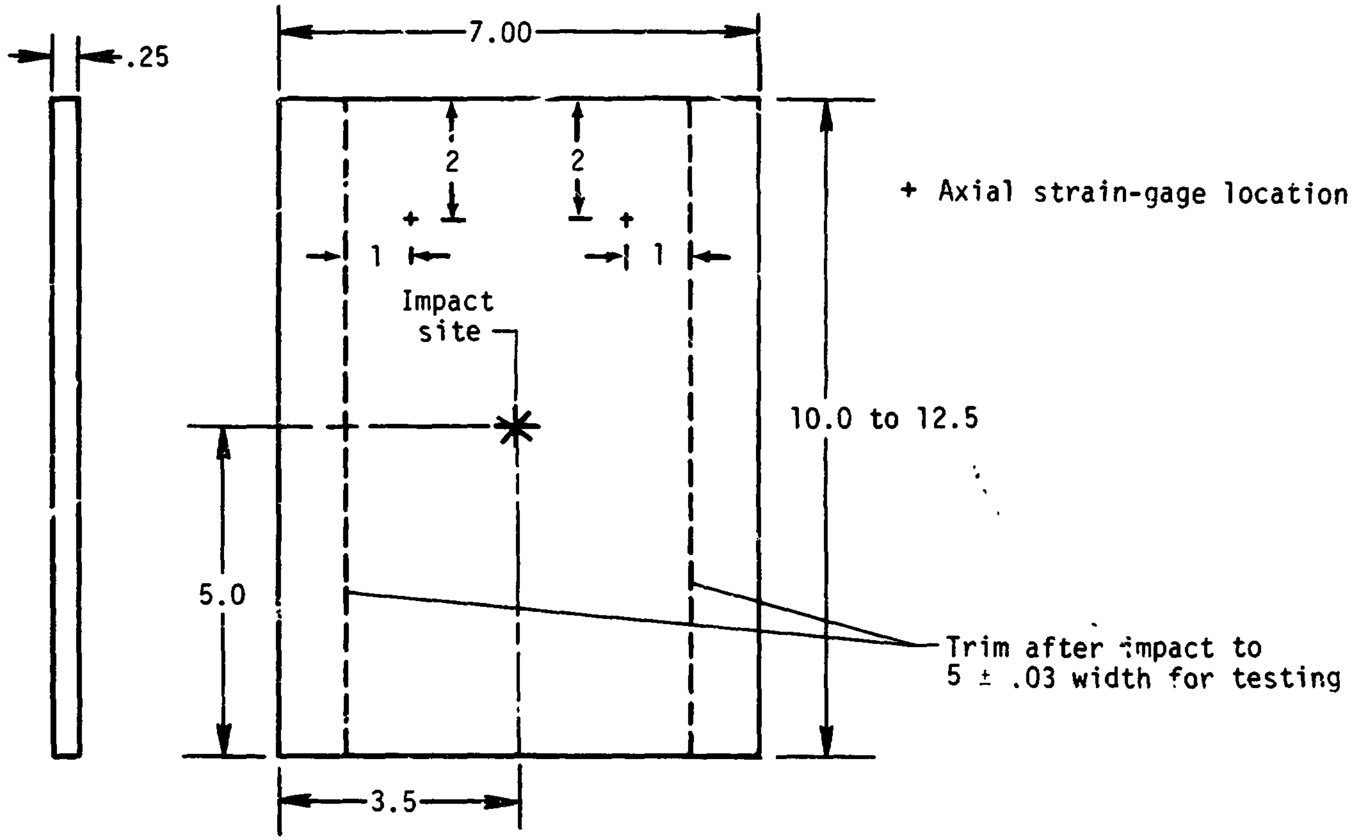

Figure 1.- Compression after impact test specimen. Dimensions are in inches.

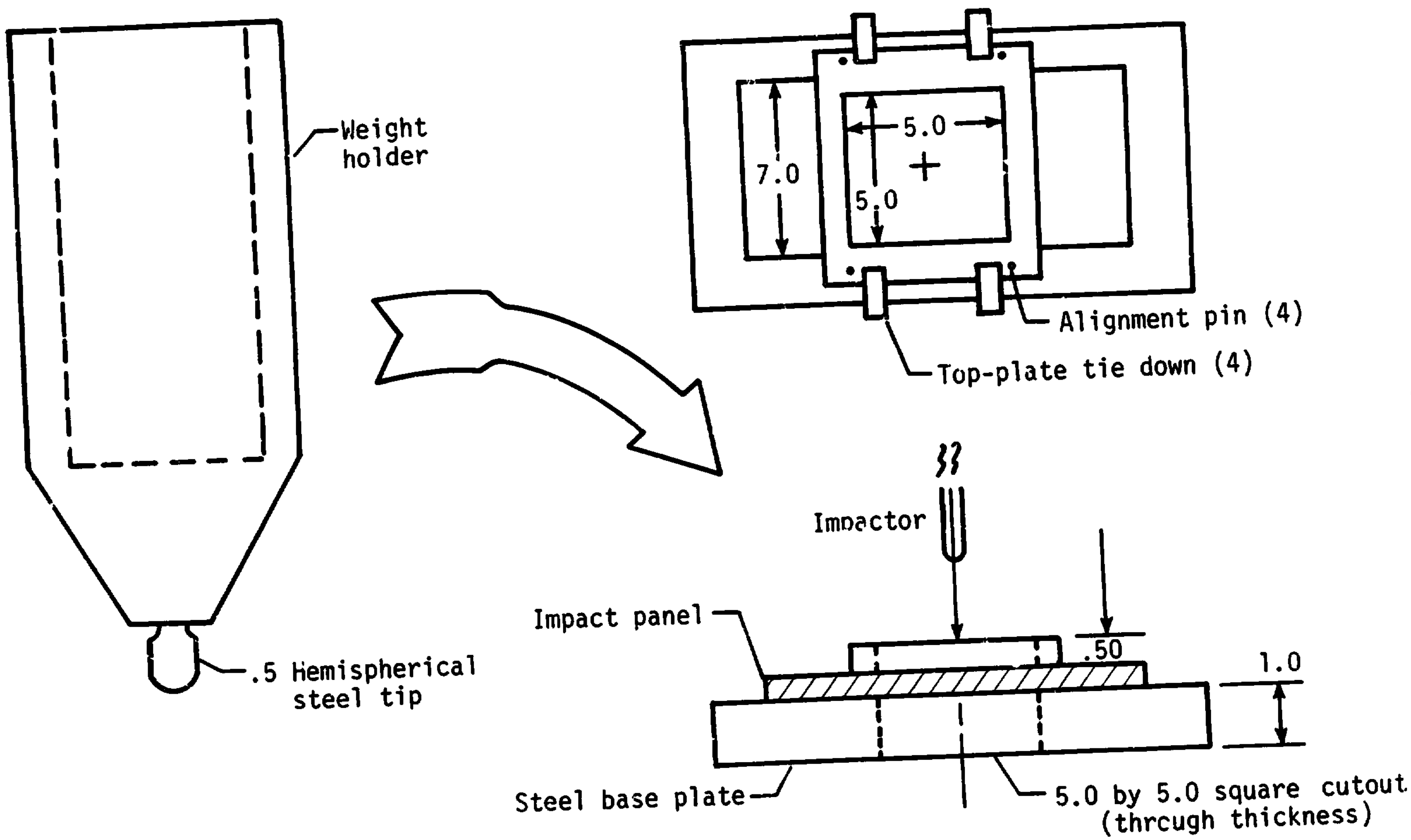

Figure 2.- Impact test apparatus. Dimensions are in inches.

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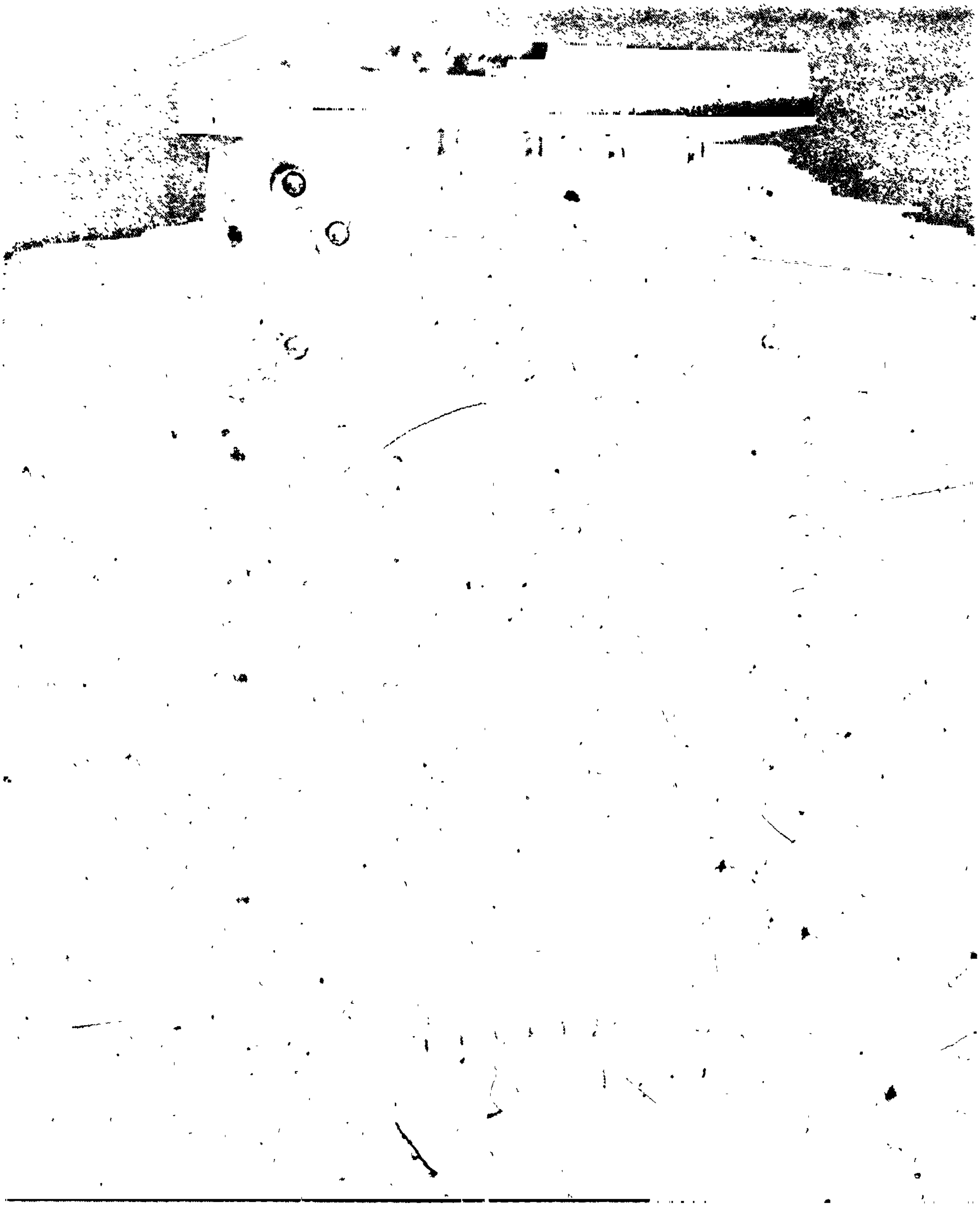

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Figure 3.- Compression test apparatus.

ST-2: SPECIFICATION FOR EDGE DELAMINATION TENSION TEST

INTRODUCTION

The ST-2 specification defines the test specimens, test apparatus, test procedures, and data to be compiled for the edge delamination tension test on graphite/epoxy laminates.

DESCRIPTION OF TEST LAMINATE

Make two panels for the chosen toughened-resin composite. One panel shall consist of an 11-ply layup $[+30/-30/+30/-30/90/90/90/-30/+30/-30/+30]_T$, and the other panel shall consist of an 8-ply layup $[+35/-35/0/90/90/0/-35/+35]_T$. Perform quality assurance C-scans on both panels and report results. Record manufacture date, batch and roll numbers, prepreg tape thickness, and basic information from vendor on fiber and resin properties. Determine and record resin content of panels after cure.

TEST SPECIMEN DIMENSIONS

Each test specimen shall be 10 in. long and 1.5 in. wide. Other details of the specimen are shown in figure 4.

TEST APPARATUS

Specimens shall be mounted in a properly aligned load frame. Use either a stroke-controlled, screw-driven machine or a stroke-controlled (or strain-controlled) hydraulic machine. (Note: "Stroke controlled" controls crosshead displacement, whereas "strain controlled" controls displacement over the gage length of the strain-measuring device mounted on the specimen.) Do not run tests in load-controlled machine. Distance between grips should be 7 in. (fig. 4). Emery cloth or tungsten carbide grit inserts should be sufficient at the grips. However, if end tabs are used, they should be squared off, not tapered (fig. 5). To measure nominal strain, use one of the two setups shown in figure 6. Mount on the specimen either (1) a pair of LVDT's (linear variable differential transducers) or DCDT's (direct-current differential transducers), one on either side, or (2) an extensometer (clip gage) with an appropriate extender arm. The gage length shall be 4 in. with gage mounts 1.5 in. from either grip (fig. 4).

TEST PROCEDURE

Five specimens of each laminate shall be tested. Prior to test, measure the laminate thickness, using micrometers, at the three locations along each edge as shown in figure 4 and record the average thickness. If the variation in thickness measurements is greater than 3 mils, record each measurement. Measure the specimen width at the three locations along the specimen length and record the average width. Load specimens at a slow rate (approximately 0.0001 in/sec). Record output of LVDT's (average of front and back) or extensometer on X-axis, and record load on Y-axis of an x-y plotter (real-time analog display). Continue loading until visible detection of edge delamination and corresponding abrupt (not continuous) deviation in load-deflection plot occurs (fig. 7). Record strain level at the onset of delamination ε_c. Note this point on the load-deflection curve. If thickness variations greater than 3 mils are found, record thickness at location closest to delamination site. Measure initial laminate modulus E_o from linear portion of load deflection curve. (See fig. 7.) For the laminate shown, the load-deflection plot is linear up to the onset of delamination. However, if gradual nonlinearity precedes the onset of delamination, measure a secant modulus E_{sec} from the origin to the delamination onset point in the load-deflection plot. This should be of concern for the 11-ply layup $[\pm 30/\pm 30/90/\overline{90}]_s$ only and not for the 8-ply layup $[\pm 35/0/90]_s$. For two of the five $[\pm 35/0/90]_s$ specimens, continue the loading until the specimen fractures into two pieces. Record the strain at failure ε_F which should be close to the vendor's reported fiber ultimate tensile strain. For all other specimens, stop loading at the onset of delamination.

TEST DATA REPORTING

For specimens tested, record (1) laminate thickness t; (2) laminate modulus E_o. and E_{sec} if necessary; (3) delamination onset strain ε_c; and (4) strain at failure ε_F for two of the $[\pm 35/0/90]_s$ specimens. Also record moduli E_{11} and E_{22}, Poisson's ratio ν_{12}, and shear modulus G_{12} for the toughened-resin composite. Describe how these properties were calculated, that is, E_{11} and ν_{12} from a $(0)_8$ tension test and G_{12} from $[\pm 45]_s$ tension test. For each edge delamination test, record the data as specified in table 2.

METHOD OF TOUGHNESS CALCULATION

Calculate $[\pm 30/\pm 30/90/\overline{90}]_s$ and $[\pm 35/0/90]_s$ laminate stiffness E_{lam} from lamina properties. Compare to average measured laminate modulus E_o. Calculate $[\pm 30]_s$ and $[\pm 35/0]_s$ laminate stiffness from lamina properties; then calculate E^* for the 11-ply layup

$$E^* = \frac{8E_{[\pm 30]_s} + 3E_{[90]}}{11}$$

and for the 8-ply layup

$$E^* = \frac{6E_{[\pm 35/0]_s} + 2E_{[90]}}{8}$$

Calculate interlaminar fracture toughness G_c for each test and then average

$$G_c = \frac{\varepsilon_c^2 t}{2}(E_{lam} - E^*)$$

where t is the average laminate thickness or the thickness measured closest to the delamination if thickness variations are greater than 3 mils. Include calculated values of all the stiffnesses and interlaminar fracture toughness G_c in table 2 where h is the thickness of one ply of the laminate.

TABLE 2.- ST-2 EDGE DELAMINATION TENSION TEST DATA

Company affiliation: ______________________

Material: ______________ h = ______ mils/ply

Laminate orientation: $[\pm 35/0/90]_s$ E_{lam} = x.xx × 10^6 psi E^* = x.xx × 10^6 psi
Laminate resin content: ___ % by weight
Test condition: 75°F dry $E_{[\pm 35/0]_s}$ = x.xx × 10^6 psi

Specimen ID	Thickness, in.	Width, in.	Delamination onset strain, µin/in.		Failure strain, µin/in.	Tensile modulus, psi	Interlaminar fracture toughness, G_c, $\frac{\text{in-lb}}{\text{in}^2}$
			(1)	(2)			
	x.xxxx	x.xxx	xxxx.	xxxx.	xxxxx.	x.xx × 10^6	x.xxx
Average:							

E_{11} = x.xx × 10^6 psi E_{22} = x.xx × 10^6 psi G_{12} = x.xx × 10^6 psi ν_{12} = 0.xxx

Laminate orientation: $[\pm 30/\pm 30/90/\overline{90}]_s$ E_{lam} = x.xx × 10^6 psi E^* = x.xx × 10^6 psi
Laminate resin content: ___ % by weight
Test condition: 75°F dry $E[\pm 30]_s$ = x.xx × 10^6 psi

Specimen ID	Thickness, in.	Width, in.	Delamination onset strain, µin/in.		Tensile modulus, psi	Secant modulus, psi	Interlaminar fracture toughness, G_c, $\frac{\text{in-lb}}{\text{in}^2}$
			(1)	(2)			
	x.xxxx	x.xxx	xxxx.	xxxx.	x.xx × 10^6	x.xx × 10^6	x.xxx
Average:							

(1) Strain at first deviation from linear stress-strain curve

(2) Strain at first visible delamination.

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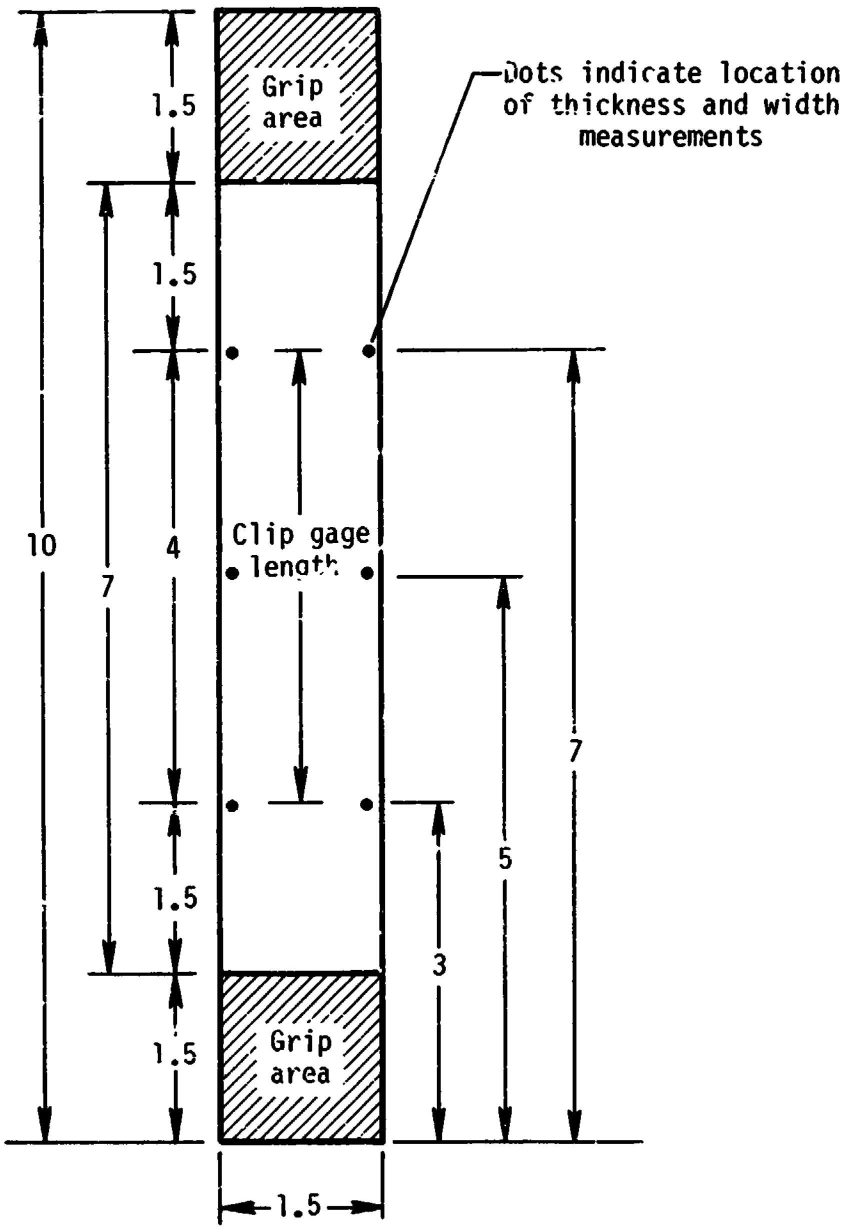

Figure 4.- Edge delamination tension test specimen.
Dimensions are in inches.

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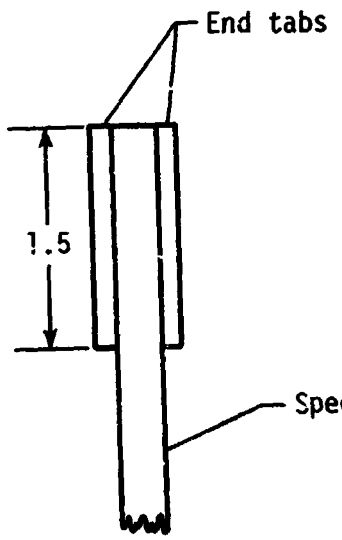

Figure 5.- Edge view of end tab configuration. Dimension is in inches.

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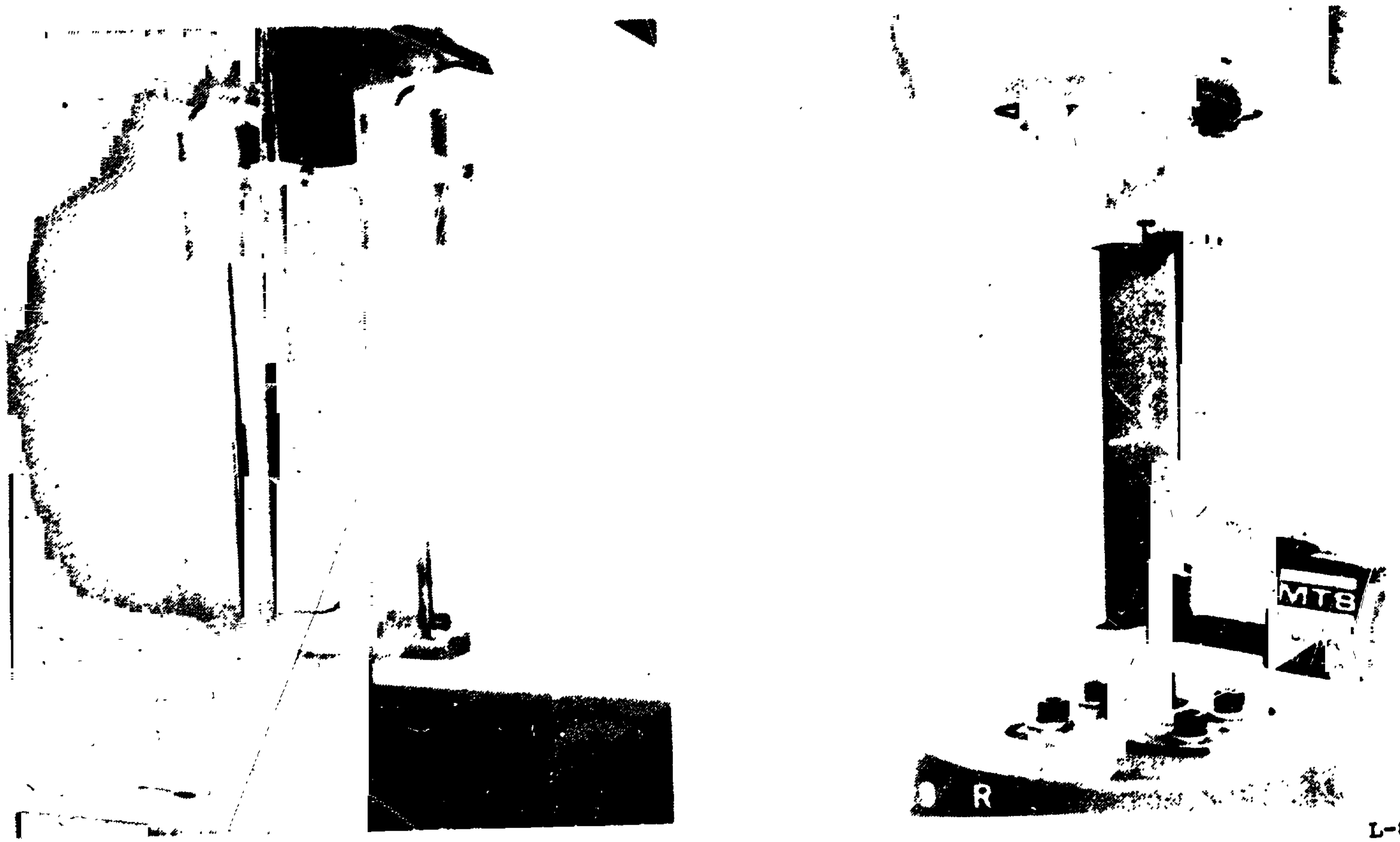

Figure 6.- Typical instrumented test specimens.

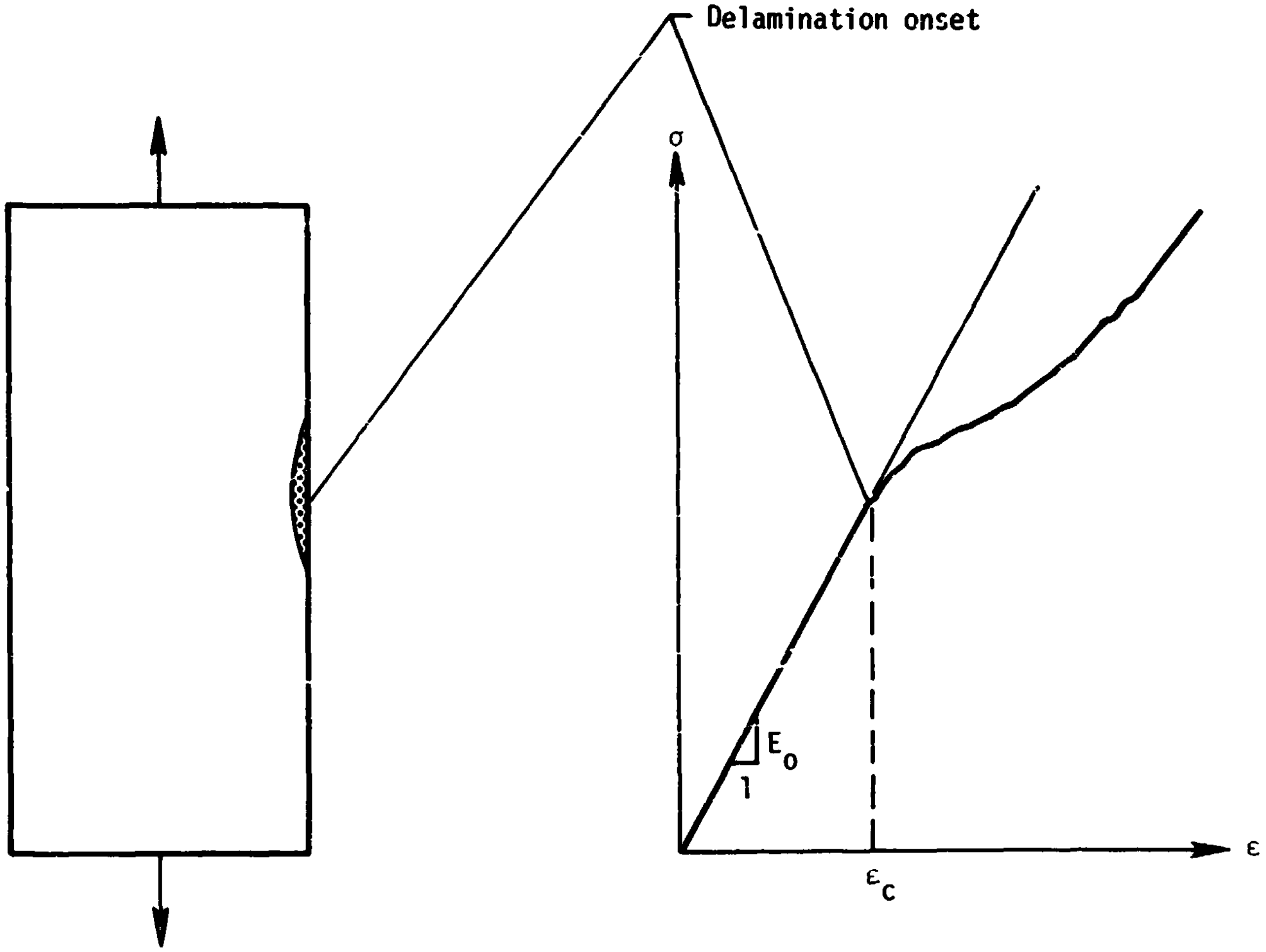

Figure 7.- Critical G_c determination.

ST-3: SPECIFICATION FOR OPEN-HOLE TENSION TEST

INTRODUCTION

The ST-3 specification defines the test specimens, test apparatus, test procedures, and data to be compiled for the ultimate tension strength testing of graphite/epoxy materials containing an open hole.

DESCRIPTION OF TEST LAMINATE

The graphite/epoxy test laminate shall have a nominal thickness of 0.25 in. and an orientation of $[+45/0/-45/90]_{ns}$. After the laminate has been manufactured, it shall be ultrasonically inspected to determine laminate quality. Record the laminate thickness and resin content. Place an identification number on one side of the laminate. Do not paint the laminate.

TEST SPECIMEN DIMENSIONS

The tension test specimen shall have a width of 2.00 in. The minimum length of the specimen shall be 12.00 in. with a minimum of 8 in. between grips. The hole diameter shall be 0.25 in. and shall be located as shown in figure 8. Each test specimen shall have one axial strain gage mounted at the location shown in figure 8. Each specimen shall have an identification number marked on it.

TEST LOADS

The test specimen shall be loaded to failure at a loading rate of approximately 20 000 lb/min. The failure strength of the specimen in figure 8 should be in the range of 15 000 to 25 000 lb.

TEST APPARATUS

Any certified tensile testing machine with a load capability >25 000 lb shall be adequate. Also, any loading fixture (such as hydraulic and "bookend") designed to properly align the specimen and preclude eccentric loading shall be acceptable.

NUMBER OF TESTS

Three tension tests shall be conducted to failure.

TEST DATA REPORTING

For each tension test, record the data as specified in table 3 where h is the thickness of one ply of laminate.

TABLE 3.- ST-3 OPEN-HOLE TENSION TEST DATA

Company affiliation: ______________________

Material: ____________ __ h = ________ mils/ply

Laminate orientation: [+45/0/-45/90]$_{ns}$ Resin content: ___% by weight Test condition: 75°F dry							
Specimen ID	Thickness, in.	Width, in.	Hole diameter, in.	Failure load, kips	Failure stress, ksi	Failure strain, μin/in.	Tensile modulus, psi
	x.xxxx	x.xxx	x.xxx	x.xx	x.xx	xxxx.	x.xx × 10^6

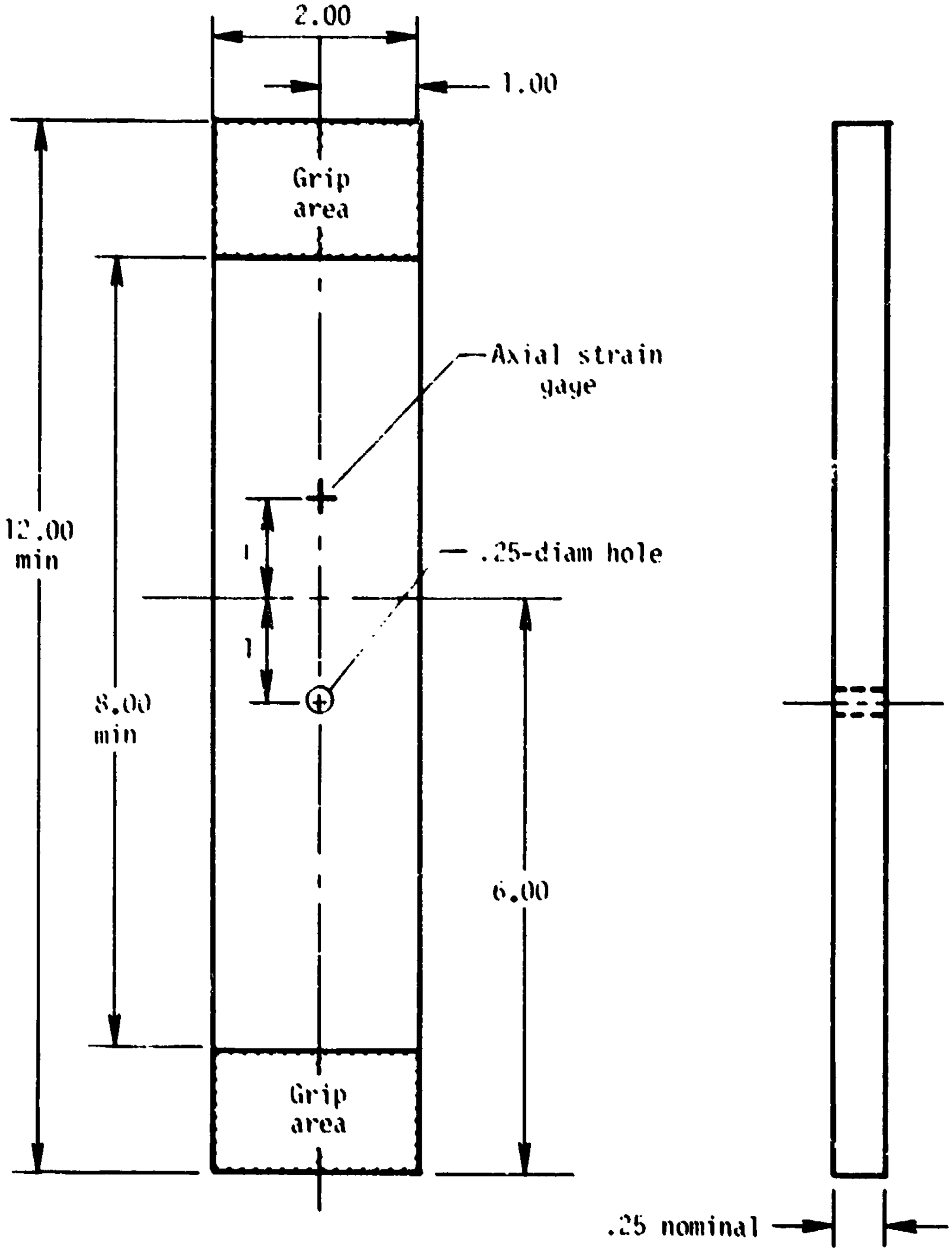

Figure 8.- Open-hole tension test specimen. Dimensions are in inches.

ST-4: SPECIFICATION FOR INPLANE OPEN-HOLE COMPRESSION TEST

INTRODUCTION

The ST-4 specification defines the test specimens, test apparatus, test procedures, and data to be compiled for inplane compression tests on graphite/epoxy laminates containing an open hole.

DESCRIPTION OF TEST LAMINATE

The graphite/epoxy test laminate shall have a nominal thickness of 0.25 in. and an orientation of $[+45/0/-45/90]_{ns}$. After the laminate has been manufactured, it shall be ultrasonically inspected to determine laminate quality. Record laminate thickness and resin content. Place an identification number on one side of the laminate. Do not paint the laminate.

TEST SPECIMEN DIMENSIONS

The compression test specimen (fig. 9) shall have a width of 5.00 ± 0.03 in. The length of the specimen shall be 10.0 in. minimum to 12.5 in. maximum. The hole diameter shall be 1.0 in. and shall be located at the center of the specimen. Each specimen shall have a minimum of four axial strain gages mounted back to back at the locations shown in figure 9. Each specimen shall have an identification number marked on it.

TEST APPARATUS

The compression test apparatus (fig. 10) shall provide simple support to the compression test specimen along its long edges oriented parallel to the compression loading direction. The short edges (loaded edge) shall be clamped between the two adjustable steel plates of the upper and lower sections of the apparatus to provide resistance to end brooming.

TEST PROCEDURE

The compression test specimen shall be loaded to failure by using a stroke-controlled testing machine. A loading rate of 0.05 in/min is recommended. The load-strain behavior of the test specimen shall be recorded throughout the test by using all four strain gages. The specimen shall be installed in the compression test apparatus (fig. 10) such that (1) the specimen is parallel to the load axis of the machine and is centered in the machine; (2) the side supports on the edges parallel to the loading axis shall be a snug fit, but not tight, so that the specimen can still slide in the vertical direction; and (3) a 0.050-in. clearance is provided between each side of the specimen and the side supports to prevent any transverse load due to Poisson deformation being restrained.

NUMBER OF TESTS

Three compression test specimens which contain a 1.0-in.-diameter open hole shall be tested.

TEST DATA REPORTING

For each compression test, record the data as specified in table 4 where h is the thickness of one ply of the laminate.

TABLE 4.- ST-4 OPEN-HOLE COMPRESSION TEST DATA

Company affiliation: ____________________

Material: ____________________ h = __________ mils/ply

Laminate orientation: $[+45/0/-45/90]_{ns}$ Resin content: ___ % by weight Test condition: 75°F dry							
Specimen ID	Thickness, in.	Width, in.	Hole diameter, in.	Failure load, kips	Failure stress, ksi	Failure strain, μin/in.	Compression modulus, psi
	x.xxxx	x.xxx	x.xxx	xx.xx	x.xx	xxxx.	$x.xx \times 10^6$

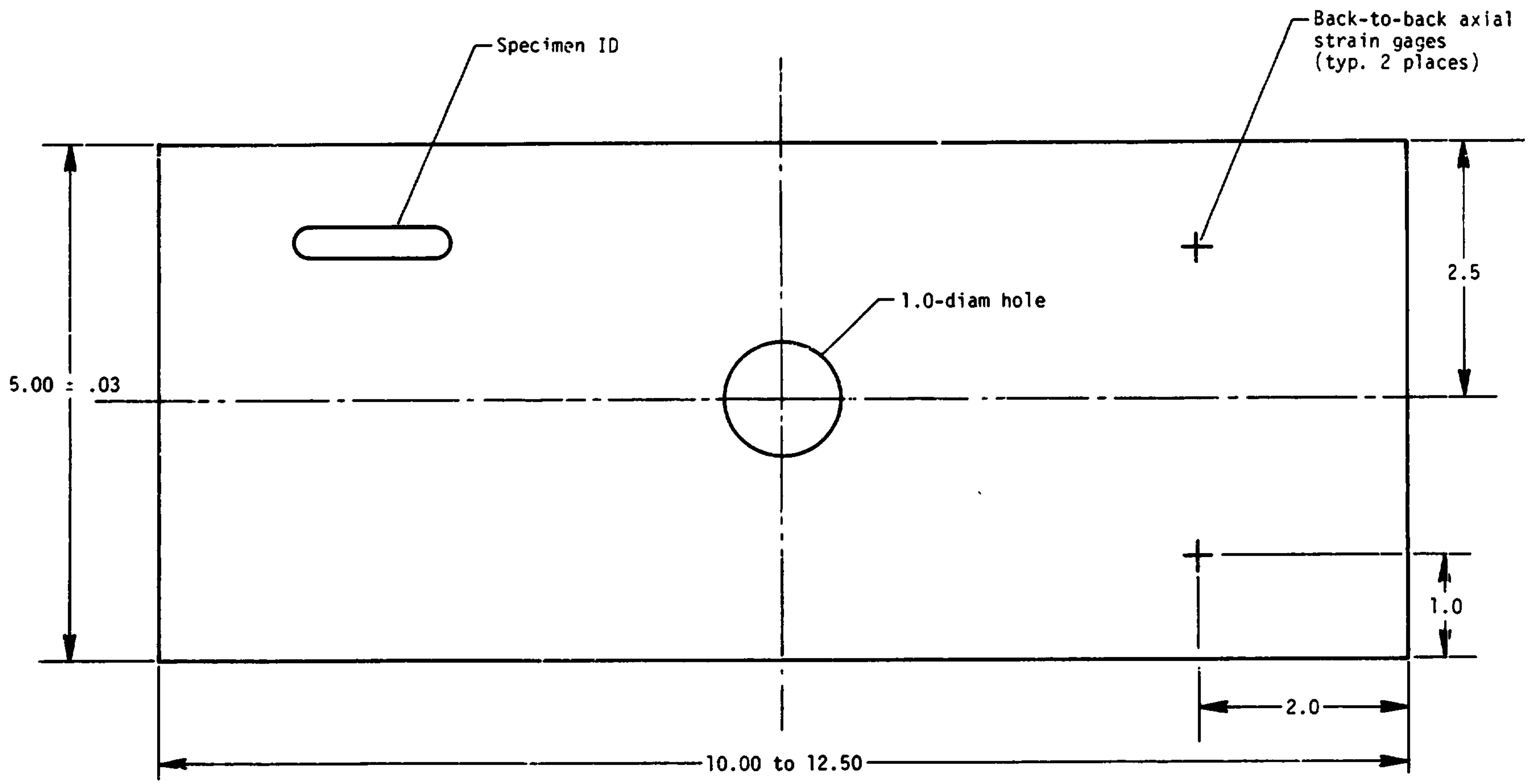

Figure 9.- Inplane open-hole compression test specimen. Dimensions are in inches.

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Figure 10.- Compression test apparatus.

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ST-5: SPECIFICATION FOR HINGED DOUBLE CANTILEVER BEAM TEST

INTRODUCTION

The ST-5 specification defines the test specimens, test apparatus, test procedures, and data to be compiled for the hinged double cantilever beam (HDCB) test.

DESCRIPTION OF TEST LAMINATE

The specimen shall have the general configuration shown in figure 11, with an even number of layers. The crack starter, 1 mil thick and made from Du Pont Teflon or Du Pont Kapton film, or equivalent, is placed between the two center layers of the laminate. For comparison purposes, it is normal to use tape material in a 0° ± 2° (longitudinal) orientation. If the material only exists in cloth form, the warp direction should be aligned parallel to the longitudinal direction of the specimen. Cloth and tape results will not be directly comparable when evaluating materials because of the transverse fibers in the cloth.

Laminate thickness should be the thinnest possible for minimum material and layup costs but thick enough to give sufficient stiffness for the two sides to act as linear elastic cantilever beams. When energy release rate G_{Ic} and flexure modulus E_f are approximately known for the material, the required laminate thickness may be evaluated from the following equation:

$$2t = 21.253(G_{Ic}/E_f)^{1/3} \tag{1}$$

In the absence of better information, it is recommended that thickness (2t) be set at approximately 0.120 in. for initial testing.

FABRICATION OF HDCB TEST SPECIMEN

Machine the laminate according to figure 11 to give an overall specimen width of 1.5 in. and a length of 9.0 in. Record laminate thickness, width, and resin content. Cut the hinge portions from extrusion MS20001, with a nominal extrusion width of 1.0 in. (from pin axis to edge). Cutting the hinge extrusion into 1.5-in. lengths prior to surface preparation is always preferred. However, for all but long-term testing in hostile environments, the hinge extrusion may be anodized or etched and primed before cutting to length. For surface preparation, anodize the hinges with phosphoric acid or chromic acid or etch according to the optimized Forest Products Laboratory (FPL) method. Prime the surface promptly with American Cyanimide BR-127 corrosion-inhibiting primer (or equivalent). Prepare the composite laminate surfaces for bonding by the following procedure. First, remove any gross contamination by cleaning with solvent (acetone or methyl ethyl ketone (MEK)). Next, remove the release agent from the caul-plate side by scrubbing with a standard household cleanser and water. Then, lightly grit blast both sides until the surface is uniformly dull in the hinge area. Rinse specimens sufficiently to remove any residual dust. Check the quality of the surface by the wetting or beading of the prepared surface. If the water beads instead of uniformly wetting the entire surface, repeat the surface preparation until the check is successful. Dry the bond surface at temperatures of 160°F to 180°F for 10 min. Adhesively bond the hinge portions with Hysol EA 9320 adhesive or an equivalent adhesive having a very high peel strength.

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Brittle adhesives with high shear strength are unsuitable for this application. The laminate and the hinges may be easily aligned by standing the specimen vertically on a flat surface with the hinged end down.

For high-temperature testing and/or for very tough materials, fasteners may be required, in addition to bonding, to increase the load-carrying capability of the hinges as shown in figure 12.

TEST APPARATUS

Any certified tensile testing machine with a displacement-controlled crosshead shall be adequate. Also, any loading fixture designed to properly align the specimen and preclude eccentric loading shall be acceptable. The test machine selected shall have a load-deflection plot capability.

TEST PROCEDURE

1. Set crosshead speed to 0.5 in/min and chart speed to 1.0 in/min. Initially set full scale to 50 lb for materials with G_{Ic} less than 2.0 in-lb/in^2.

2. Mount specimen in machine by gripping upstanding legs of hinge.

3. Apply load of approximately 10 lb and set chart to agree with this reading.

4. Start chart and increase load until crack length (measured from center of hinge pins to crack tip) reaches approximately 2.0 in. Stop machine.

5. Mark the crack tip on both edges of the specimen to allow crack length to be measured at a later stage. The average of the lengths from the two sides will be designated as "a_1."

6. Mark the chart so that this point in the test can be identified after the test is completed.

7. Restart machine and extend crack length to approximately 3.0 in. Stop the machine, mark crack tips, and identify point on chart as before.

8. Repeat procedure for crack lengths of approximately 4.0, 5.0, and 6.0 in. Run machine to extend crack tip just beyond final stop point.

9. Unload specimen and remove from testing machine.

NUMBER OF TESTS

Three or more HDCB tests shall be conducted for each material.

TEST DATA REPORTING

Loads (P_1, P_2, P_3, ...), deflections (δ_1, δ_2, δ_3, ...), and average crack lengths (a_1, a_2, a_3, ...) shall be measured as shown in figure 13 and described in the test procedure and recorded as specified in table 5 where h is the thickness of one ply of the laminate.

STRAIN-ENERGY RELEASE RATE CALCULATION

The strain-energy release rate shall be calculated by two methods: (1) modified direct beam equation method and (2) energy-area integration method. These two methods are described in this section.

Nomenclature

A	area of crack surface
a	crack length
a_o, E_o	constants used for curve fit
b	width of beam
C	compliance of double cantilever beam
E_{af}	apparent flexure modulus
G_{Ic}	critical strain energy release rate
P	load
t	thickness of beam
U	strain energy
β	correction factor for double cantilever beam equation
δ	deflection of double cantilever beam

Modified Direct Beam Equation Method

An expression for G_{Ic} for the hinged double cantilever beam (HDCB) specimen is related to the compliance and load by[1]

$$G_{Ic} = \frac{P^2}{2b}\left(\frac{dC}{da}\right) \tag{2}$$

where dC/da is the rate of change of compliance with crack length. The compliance at the loading point is the displacement per unit load

$$C = \frac{\delta}{P} \tag{3}$$

Substituting equation (3) into equation (2) gives

$$G_{Ic} = \frac{P\delta}{2bC}\left(\frac{dC}{da}\right) \tag{4}$$

The compliance of the elastic beam is

$$C = \frac{8a^3}{E_{af}bt^3} \tag{5}$$

where E_{af} is a function of crack length a. Taking the derivative of compliance with respect to crack length gives

$$C' = \frac{dC}{da} = \frac{24a^2}{E_{af}bt^3}\left[1 - \left(\frac{aE'_{af}}{3E_{af}}\right)\right] \tag{6}$$

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[1]Paris, Paul C.; and Sih, George C.: Stress Analysis of Cracks. Fracture Toughness Testing and Its Applications, Spec. Tech. Publ. No. 381, American Soc. Testing & Mater., c.1965, pp. 30-81.

Substituting equations (5) and (6) into (4) gives

$$G_{Ic} = \frac{3P}{2b}\left(\frac{\delta}{a}\right)\beta \tag{7}$$

where

$$\beta = \left(1 - \frac{aE'_{af}}{3E_{af}}\right) \tag{8}$$

The factor β can be calculated from the experimentally fitted apparent flexure modulus expression. For composites, Ripling and Santner[2] have shown that the best fit was obtained with the following expression:

$$E_{af} = E_o\left(1 - \frac{a_o}{a}\right) \tag{9}$$

where E_o and a_o are the constants evaluated by the method of least squares. The least-squares equations are

$$\left.\begin{aligned} nE_o + a_oE_o \sum_{i=1}^{n} - \left(\frac{1}{a_i}\right) &= \sum_{i=1}^{n} \left(E_{af}\right)_i \\ E_o \sum_{i=1}^{n} - \left(\frac{1}{a_i}\right) + a_oE_o \sum_{i=1}^{n} \left(-\frac{1}{a_i}\right)^2 &= \sum_{i=1}^{n} \left(E_{af}\right)_i \left(-\frac{1}{a_i}\right) \end{aligned}\right\} \tag{10}$$

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[2]Work done by E. J. Ripling and J. S. Santner, of Materials Research Laboratory, Inc., under Contract NAS3-21824 for Lewis Research Center.

In order to make the expression for β a function of crack length a only, differentiate equation (9) to get

$$E'_{af} = \frac{d\left(E_{af}\right)}{da} = \frac{E_o a_o}{a^2} \tag{11}$$

Substituting equations (9) and (11) into equation (8) gives

$$\beta = \frac{3a - 4a_o}{3(a - a_o)} \tag{12}$$

The final expression for the modified direct (because all the variables can be measured directly from the HDCB test) beam equation is

$$G_{Ic} = \frac{P\delta}{2ab}\left(\frac{3a - 4a_o}{a - a_o}\right) \tag{13}$$

Energy-Area Integration Method

By definition, the strain-energy release rate is equal to the energy required to extend a preexisting crack an infinitesimal unit of area:

$$G_{Ic} = \frac{dU}{dA} \tag{14}$$

The HDCB test provides a plot of force P at the hinges versus deflection δ at the hinges. When the force is integrated over the deflection, the result is energy. To compute the energy required to extend a crack from a_1 to a_2, three separate energies are considered:

(a) The energy stored in the beam prior to the crack propagation at a_1, as shown in figure 14(a)

(b) The additional energy required to propagate the crack from a_1 to a_2 by further flexing the beam by increasing the deflection, as shown in figure 14(b)

(c) The energy remaining in the beam after the crack propagation is halted at the crack length of a_2, as shown in figure 14(c)

The first two energies ((a) + (b)) minus the third ((c)) is the total energy required to propagate the crack from a_1 to a_2, as shown in figure 14(d). The strain-energy release rate is this energy divided by the area created by the crack extension from a_1 to a_2. Thus, the measurements required to calculate G_{Ic} by this method are force-deflection plot, initial and final crack lengths, and specimen width. The freedom from measuring the crack lengths during testing greatly facilitates the testing in an environmental chamber.

The modified direct beam method calculates G_{Ic} at the instant of crack propagation, whereas the energy-area integration method calculates G_{Ic} for the entire fractured area a_1 to a_2, which includes peaks for the crack initiation point and valleys for the crack arrest point. For this reason, the energy-area integration method gives slightly lower G_{Ic} values than those obtained from the modified direct method.

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TABLE 5.- ST-5 HINGED DOUBLE CANTILEVER BEAM TEST DATA

Company affiliation: ____________________

Material: ____________ h = ______ mils/ply

Laminate orientation: $[0]_n$ Resin content: ___ % by weight Test condition: 75°F dry																		
Coupon ID	Total thickness, 2t, in.	Width, in.	a_1, in.	δ_1, in.	P_1, lb	a_2, in.	δ_2, in.	P_2, lb	a_3, in.	δ_3, in.	P_3, lb	a_4, in.	δ_4, in.	P_4, lb	a_5, in.	δ_5, in.	P_5, lb	G_{Ic}, $\frac{\text{in-lb}}{\text{in}^2}$
	x.xxxx	x.xxx	x.xx	x.xx	xxxxx.													x.xxx

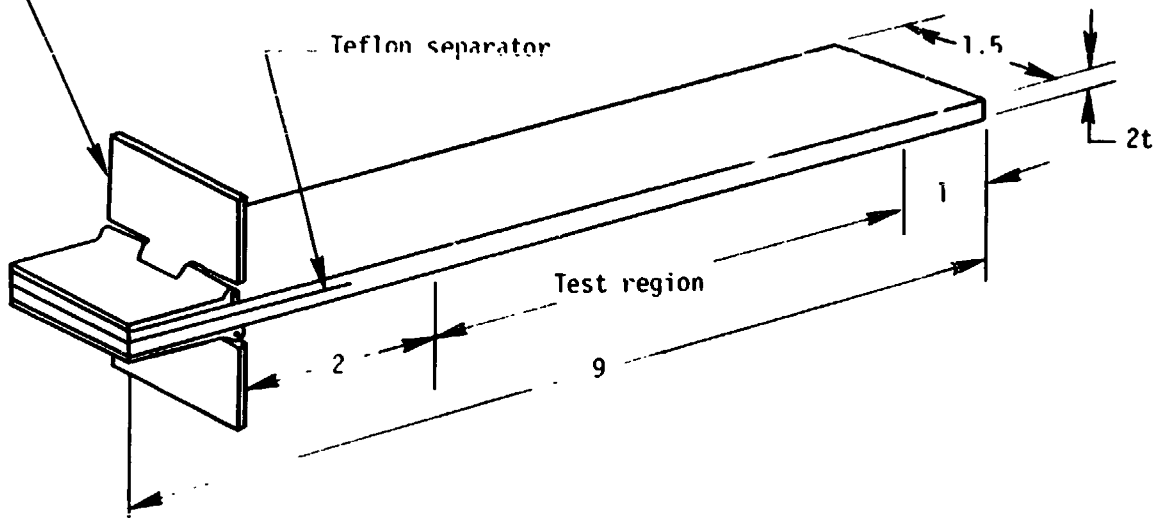

Figure 11.- Hinged double cantilever beam (HDCB) specimen.
Dimensions are in inches.

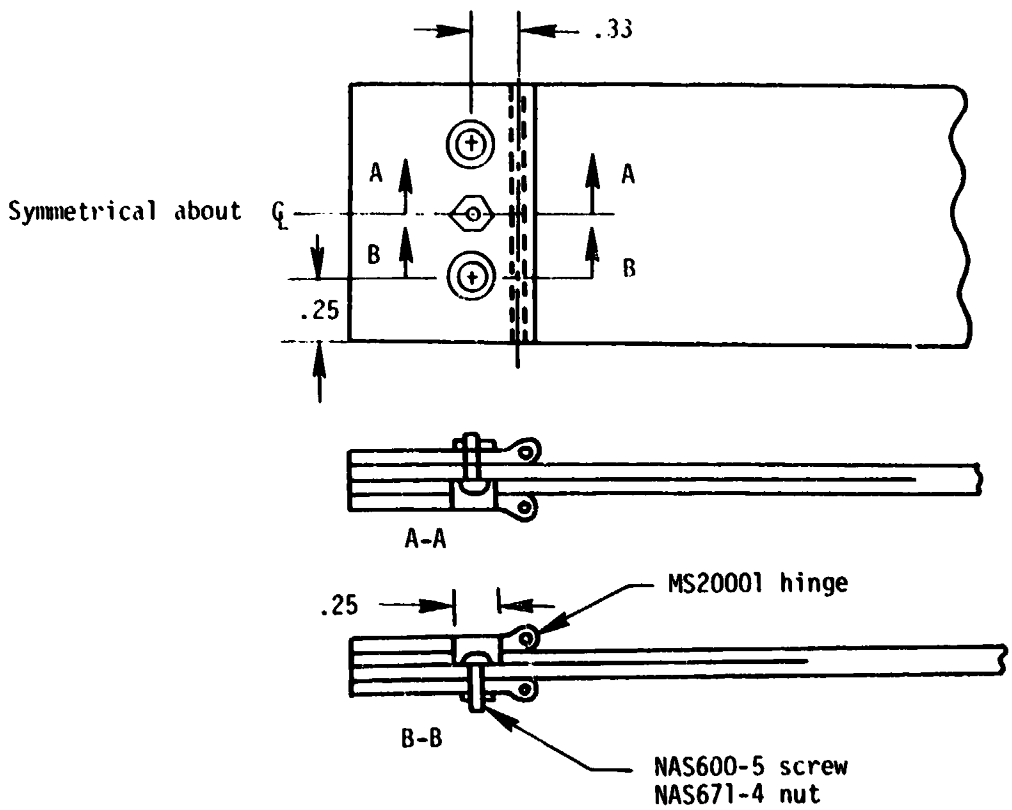

Figure 12.- Increased load-carrying capability by fasteners.

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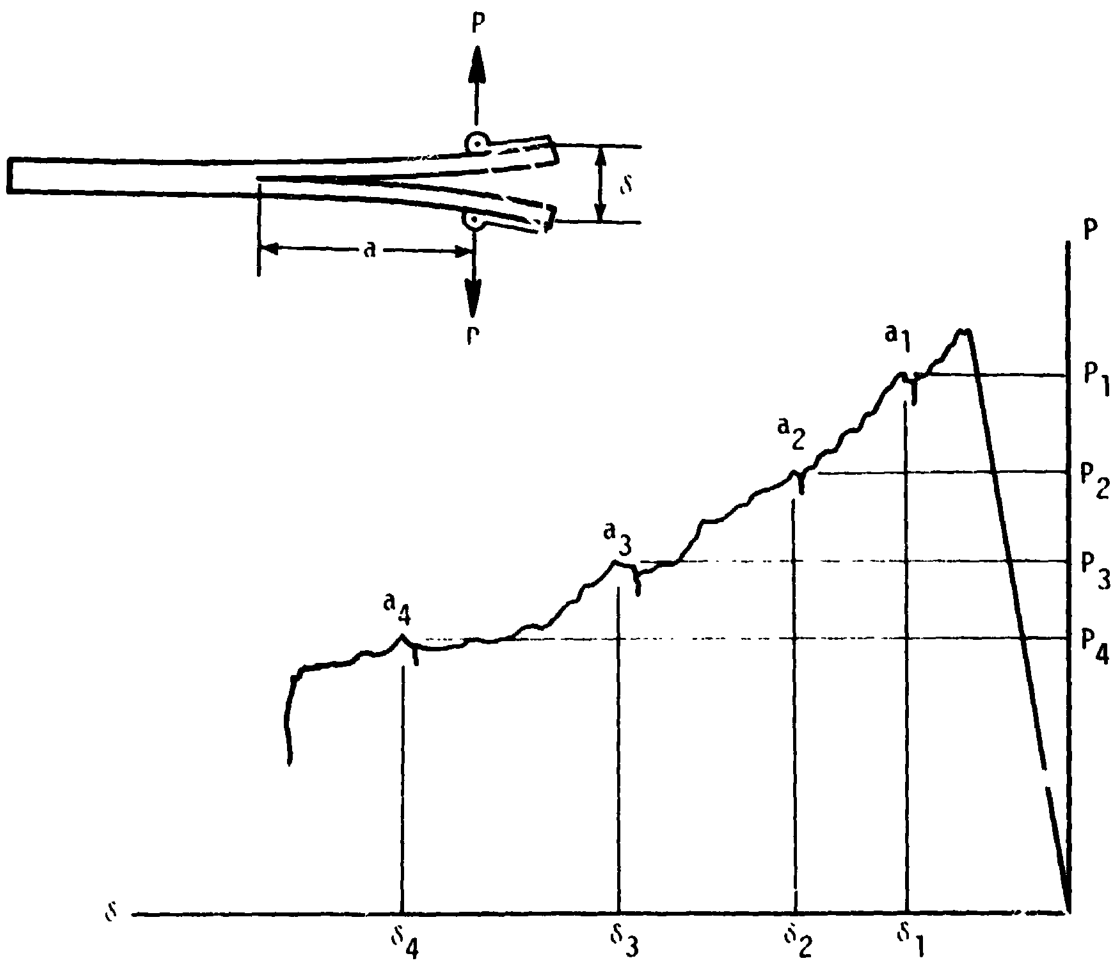
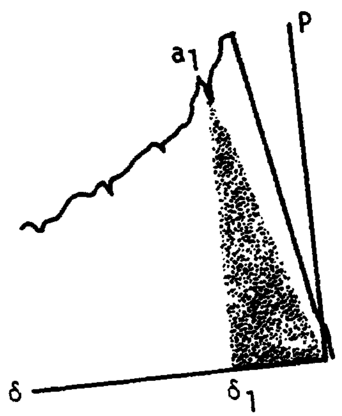
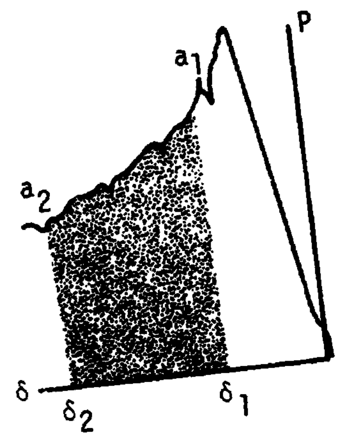


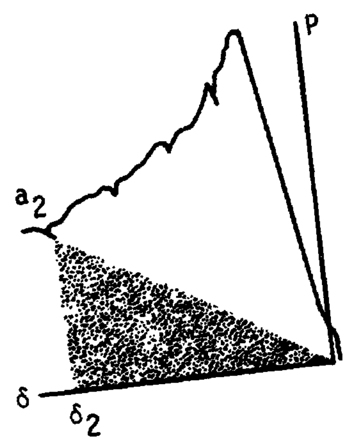
Figure 13.- Loads and deflections for specified crack length.



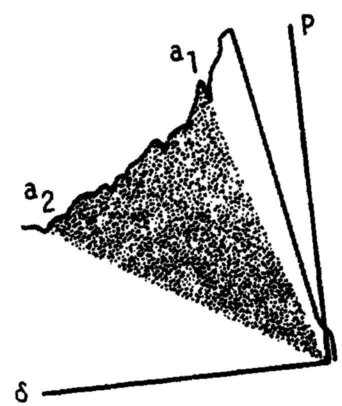
(a) Energy to flex beam to crack initiation.



(b) Additional energy to propagate crack and flex beam further.



(c) Energy remaining in flexed beam after crack propagation.



(d) Total energy to propagate crack ((a) + (b) - (c)).

Figure 14.- Energy-area integration method for calculating G_{Ic}.